HSCT WING DESIGN TROUGH MULTILEVEL DECOMPOSITION

Peter J. Röhl*, Dimitri N. Mavris**, Daniel P. Schrage†

School of Aerospace Engineering, Georgia Institute of Technology
Atlanta, GA 30332-0150

Abstract

A multilevel decomposition approach for the preliminary design of a High Speed Civil Transport Aircraft wing structure is described. The wing design is decomposed into three levels. The top level uses the FLOPS aircraft synthesis program to generate preliminary weights, mission, and performance information. The optimization criterion is productivity expressed by a productivity index for the specified mission. The second level of the system performs a finite-element based structural optimization of the wing box with the help of the ASTROS structural optimization tool. The wing structure is sized subject to strength, buckling, and aeroelastic constraints. The buckling constraint information is supplied by the third level where a detailed buckling optimization of individual skin cover panels is performed. The process is then verified with the help of data from supersonic transport studies performed by US aerospace companies in the 70s. Finally, an HSCT configuration based on the NASA HiSAIR H 24 e is optimized using the multilevel decomposition scheme. The gross weight is reduced by 9.5 %, and the productivity index, the system level objective function, is increased by 15 % for the most promising of the configurations analyzed.

Nomenclature

a_{ij}	shape function coefficient
PI	productivity index
t	panel thickness
V_b	block speed
W_e	empty weight
W_f	fuel weight
W_p	payload
x	design variable vector
ξ	nondimensional chordwise coordinate
η	nondimensional spanwise coordinate

* Research Associate, Member AIAA
** Research Engineer, Member AIAA
† Professor, Aerospace Engineering, Member AIAA

Introduction

As modern aircraft designs tend to become more and more complex in order to outperform previous models, new techniques in system design synthesis and optimization become increasingly important. This is especially true for the design of a second - generation supersonic transport aircraft as an example of a highly coupled system. At the same time, the methodology of multidisciplinary design and optimization is evolving into a new engineering discipline that seems most suitable to address this type of design problem where the traditional sequential approach will most likely lead to suboptimal results[1].

One obstacle for the fast evaluation of a relatively large number of candidate configurations in the development of a High-Speed Civil Transport Aircraft (HSCT) has been the long time, up to 24 months, for the completion of one full design cycle[2]. At the same time, studies performed in the 70s indicate that a sequential addressing of the strength and flutter problem in the structural design of a supersonic transport wing leads to severe mass penalties[3].

All these factors combined clearly show the need for an integrated wing design procedure that is able to address structural design, aerodynamic, and aeroelastic questions early in the design process. The three-level wing design procedure presented in this paper can be regarded as a framework where additional modules, for example controls, more accurate aerodynamics, propulsion, etc. can be integrated at a later stage. The material presented here is based on ongoing efforts at Georgia Tech to develop, enhance, and implement the technologies of Concurrent Engineering (CE) and Integrated Product and Process Design (IPPD)[4,5].

Multilevel Decomposition Procedure

Decomposition of the Design Task

The wing structural design problem is decomposed into three levels in a hierarchical structure (Fig. 1). At the top level, a general aircraft sizing and performance code sizes the aircraft for the specified mission based on statistical, empirical, and analytical methods. At the middle level the actual structural layout of the wing takes place based on a relatively crude finite element analysis. On the third level individual skin cover panels which are modeled as membrane elements with a smeared thickness at the second level are sized for buckling as stiffened panels.

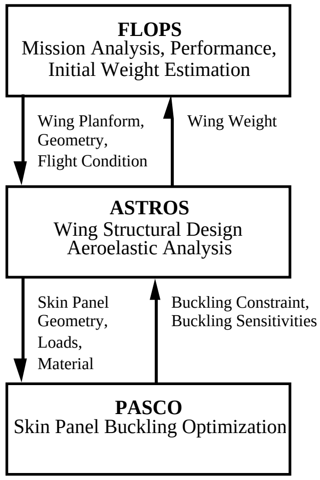

Fig. 1: Multilevel Decomposition of the Wing Design Problem

Analysis and Design Modules

The top level uses the code FLOPS (Flight Optimization System)[6] developed by NASA Langley which has been modified for this application and for its integration into the multilevel scheme. FLOPS is a multidisciplinary system of computer programs for conceptual and preliminary design and evaluation of advanced aircraft concepts. It consists of nine primary modules for weights, aerodynamics, engine cycle analysis, propulsion data scaling and interpolation, mission performance, takeoff and landing, noise footprint calculation, cost analysis, and program control. The weights module uses statistical / empirical equations to predict the weight of each item in a group weight statement and also calculates centers of gravity and moments of inertia. The aerodynamics module provides drag polars for performance calculations. The engine cycle analysis module provides the capability to internally generate an engine deck consisting of thrust and fuel flow data at a variety of Mach-altitude conditions. The mission performance module uses the calculated weights, aerodynamics, and propulsion data to calculate performance and the fuel balance. Through the program control module, FLOPS may be used to analyze a point design, parametrically vary certain design variables, or optimize a configuration with respect to these design variables (such as minimum gross weight, maximum range, minimum cost, etc.). The configuration design variables include wing area, wing sweep, wing aspect ratio, wing taper ratio, wing thickness to chord ratio, gross weight, and thrust. The performance design variables are cruise Mach number and maximum cruise altitude. The engine cycle design variables are the design point turbine entry temperature, the maximum turbine entry temperature, the fan pressure ratio, the overall pressure ratio, and the bypass ratio for turbofan and turbine bypass engines.

The Productivity Index PI, defined as the ratio of aircraft productivity to the sum of fuel and empty weight,

$$PI = \frac{W_p \cdot V_B}{W_e + W_f} \ , \tag{1}$$

has been selected as a measure of aircraft performance and has been programmed as a possible objective function. At a time when economic data for a supersonic transport aircraft are sketchy at best, the productivity index offers a measure of comparing different configurations by normalizing aircraft productivity (block speed times payload) with respect to an indicator of the cost involved in achieving this productivity. The denominator captures a part of both the operating costs (through the fuel weight which directly translates into fuel cost) and the acquisition cost which is usually calculated as a function of aircraft empty weight.

The structural optimization level uses the Automated Structural Optimization (ASTROS[7]) code to design a minimum weight wing subject to a large number of stress, strain, displacement and flutter constraints. ASTROS is a multidisciplinary analysis and design tool most suitable for the design of aerospace structures. It was developed for and by the Flight Dynamics Laboratory, Air Force Wright Aeronautical Laboratories, and has been continuously upgraded. The latest version being used now is Version 11. It combines finite-element-based structural analysis, aerodynamic and aeroelastic analysis with mathematical optimization algorithms in order to design a minimum weight structure meeting a variety of different types of constraints. The engineering analysis capabilities include both static and dynamic structural analyses (transient and steady-state) and static and dynamic aeroelastic capabilities. Design constraints include stress, strain, displacement, frequency, flutter, and aerodynamic constraints. Data storage and manipulation is performed by ASTROS's own database system (CADDB). Steady aerodynamic analyses in ASTROS are performed by the USSAERO code, while the Doublet-Lattice and constant pressure methods are used for unsteady analyses in the subsonic and the supersonic regime, respectively.

The standard ASTROS solution sequence has been modified to allow a stop and restart of the optimization procedure after a certain number of iterations in order to allow the designer to review the design progress and to facilitate the call to the panel buckling analysis on the third level of the multilevel decomposition scheme.

The wing structure is modeled consisting of spars, ribs, and skin panels. The skin panels are modeled as membrane elements, the spar webs and the ribs as shear panels, and the spar caps as rod elements (Fig. 2). All these elements can be designed, whereas posts that connect the upper and lower wing surface are modeled as rod elements that are not designed and mainly serve the purpose of preventing the upper and lower surface from collapsing onto each other.

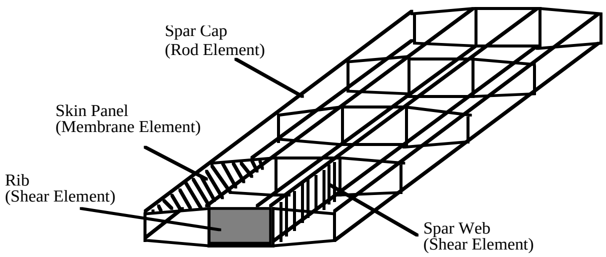

Fig. 2: Wing Box Finite Element Model

The number of designed elements for the HSCT wing ranges from a few hundred to over 1000, depending on the number of ribs and spars in the wing. Therefore, design variable linking schemes are necessary to reduce the number of design variables to a number that the optimizer can manage. ASTROS offers basically two ways of design variable linking, physical linking where the designed property of a selection of elements of the same type is set to one design variable, and shape function linking. In the case of shape function linking, the design variables are the coefficients of a polynomial, and the value of the polynomial at a certain location determines the value of the designed property of that specific element. In this case, a two-dimensional shape function of the form

$$t = a_{00} + a_{10}\xi + a_{01}\eta + a_{11}\xi\eta + a_{20}\xi^2 + a_{02}\eta^2 \qquad (2)$$

is being used to model spar caps, webs, and skin panels, whereas the rib panels are physically linked to one design variable for each rib. One-dimensional shape function linking which proves beneficial especially when the number of spars is small can be achieved by setting the corresponding coefficients in the other direction to zero. The wing is subdivided into three design regions (Fig. 3), in each of which the design variable linking scheme can be selected individually.

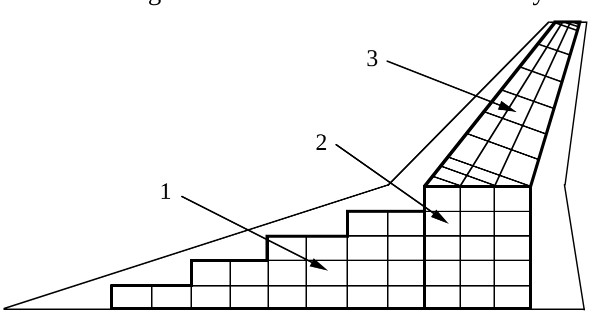

Fig. 3: Wing Design Regions

The component level of the three-level procedure optimizes selected wing skin panels for buckling. It uses the code PASCO (Panel Analysis and Sizing Code)[8,9] developed by NASA Langley. PASCO was developed for the buckling and vibration analysis and sizing of prismatic structures having an arbitrary cross section. PASCO is primarily intended for analysis and sizing of stiffened panels made of laminated orthotropic materials. When used in the analysis mode, PASCO calculates laminate stiffnesses, laminate stresses and strains, buckling loads, vibration frequencies, and overall panel stiffness. When used in the sizing mode, PASCO adjusts sizing variables to provide a low-mass panel design that carries a set of specified loadings without exceeding buckling or material strength allowables.

The details of the integration of PASCO and ASTROS and the formulation of a panel buckling constraint in ASTROS were subject of a previous publication[10] and are therefore not discussed here.

Interfaces

A finite element pre-processor specifically designed for this problem has been written that takes the FLOPS wing geometry output and places a wing box into this geometry complete with all grid points, elements, element connectivity, static airloads, and design variable definition and linking scheme. The pre-processor creates the complete ASTROS input file with the help of a small skeleton file that mainly contains the ASTROS solution control commands. The user selects how he wants to model the wing structure (number of ribs, spars, design variable linking, number of different wing sections, initial and minimum values for the design variables) as described above. The fuel weight determined in FLOPS is distributed as point masses onto the inboard grid points for vibration and flutter analysis. The engine attachment to the wing structure is modeled by connecting rod elements that automatically connect the engines to the lower surface grid points closest to the spanwise engine locations in FLOPS. For static analyses with a cantilevered boundary condition, each engine is modeled by two point forces on the attachment structure, and for dynamic and "free-free" analyses, the engine masses are modeled as rods containing non-structural mass distributed along their length. Since the engine weights and locations are the actual values taken from the FLOPS file, the influence of engine placement on wing dynamic and aeroelastic behavior can be investigated.

The PASCO pre-processor accesses the ASTROS database and reads the information necessary to model user-selected skin panels, i.e. panel geometry, shape function design variable values, material constants and allowables. Out of this information a PASCO input file is automatically created that models the membrane element from ASTROS as a uniaxially stiffened panel. Different stiffener types are possible (blade, hat, Z, T, etc.).

The PASCO post-processor takes the load at which the individual optimized skin panel starts to buckle, together with its derivatives with respect to the design variables, and places these values into the ASTROS database as the basis to formulate the ASTROS buckling constraint and its sensitivities to be used for the next ASTROS iteration.

Overall System Control and Execution

The multilevel design procedure is controlled and executed by UNIX C-shell scripts, some of which also perform the data filtering tasks between the individual codes. The user interacts with the main driver in order to control the flow of the execution and to issue interrupt or restart commands. The designer would typically start with a FLOPS analysis that creates a baseline design for the prescribed mission. With this information, the finite element preprocessor is run to create the ASTROS input file. Then the ASTROS - PASCO procedure can be started and run for a few - for example four or five - iterations. After reviewing the design, the designer can choose whether he wants to continue with ASTROS-PASCO or whether the current wing weight is so far off the FLOPS assumption that it seems necessary to rerun FLOPS. He can also decide to start the system level optimization at this point. Since the system level optimization is by orders of magnitude faster than the ASTROS - PASCO procedure, it proves beneficial to stop the latter after only a few iterations - before convergence is reached - and start the system level optimization. The system level optimization can be run until convergence is reached since it is very fast and the time savings by stopping it prematurely because of the fact that the lower level has not converged yet are negligible. The ASTROS-PASCO procedure can then be started again with the new - optimized - planform, twist and camber, and weights. The procedure has the capability of using the final design variable values of the previous iteration as the new starting point so that it does not have to start from scratch again. The design variable values are automatically extracted from the ASTROS database and placed into the ASTROS input file.

Experience shows that the influence of the design weights of the aircraft on the structural weight of the wing are small compared to the influence of geometry, so that as long as planform changes are small, this procedure is highly convergent; three or four iterations between the system level and the structural optimization are usually sufficient. For the time being, no fixed convergence criterion has been programmed into the overall procedure. It is basically up to the designer to make the decision where he considers the procedure converged.

A simple parallel processing capability for the skin panel buckling optimizations has been implemented. For each iteration of the structural optimization a number of skin panels are identified as buckling-critical, and, therefore, each of these panels has to be optimized. All these panel optimizations are totally independent of each other and can be executed in parallel. Currently the procedure is implemented on a cluster of seven IBM RS-6000 workstations, and all panel optimization cases are distributed onto these workstations and executed as remote shells. The driver shell script simply pauses until all panel optimizations have terminated and then collects the results and places them into the system database. In this fashion the overall execution time of the procedure can be considerably reduced.

Verification of the Procedure

Two supersonic transport configurations were used as test cases for the procedure, the BAC-Sudaviation Concorde as the only supersonic transport in operation today, and a study performed by Lockheed California, concluded in 1977[3], which was mentioned before. Due to space limitations, only the results obtained with the Lockheed SST configuration are described here. Ref. 11 contains detailed results for both aircraft.

The Lockheed SST Study

The final configuration of the Lockheed SST studies was a Mach 2.7, 234 passenger aircraft with a design range of 4200 nm and a take-off gross weight of 750,000 lb. The airplane was propelled by four duct-burning turbofan engines with a sea level static thrust of 89,500 lb each. Fig. 4 shows a drawing of the final configuration.

A system level optimization with the productivity index as the objective function was not considered useful since the results would probably not lead to new insights. The Lockheed SST does not necessarily represent a PI-optimum so that it would not be likely to obtain the "correct" geometry in a system level optimization. Therefore, the system level synthesis was only executed in its analysis mode, trying to match the empty weight and the fuel weight for the required mission. This proved only to be possible by considerably increasing the wing weight which was by far underpredicted by FLOPS, one of the main reasons for the creation of this procedure. Table 1 shows the weight breakdown of the Lockheed SST in comparison with the FLOPS results. The question to be answered now is whether it is possible to design a wing structure subject to its design loads so that the total wing weight matches this value.

Fig. 4: The Lockheed SST

		Lockheed	FLOPS
Performance, General			
Passengers		234	234
Range	[nm]	4200	4200
Cruise Mach No.		2.7	2.7
Max. Cruise Altitude	[ft]	70000	70000
Productivity Index	[Kts]		87.35
Engine Type		Duct-Burning Turbofan	Duct-Burning Turbofan
Net Thrust (SLS)	[lb]	89500	89500
Number of Engines		4	4
Geometric Data			
Fuselage Length	[ft]	287.0	287.0
Wing Span	[ft]	132.5	132.5
Wing Area	[ft^2]	10923.0	10874.5
Aspect Ratio		1.607	1.612
LE Sweep, Inboard	[deg.]	74.00/70.84	75.32
LE Sweep, Outboard	[deg.]	60.00	60.00
Wing T/CRatio	[%]	2.65	2.65
Weights			
Wing	[lb]	90584.	88287.
Fuselage	[lb]	42122.	50245.
Total Structure	[lb]	187722.	188335.
Propulsion	[lb]	70884.	57874.
Syst. and Equipm.	[lb]	41680.	60464.
Empty Weight	[lb]	303144.	306672.
Operating Items	[lb]	10700.	9252.
Oper. Empty Wght	[lb]	313844.	315924.
Passengers	[lb]	39000.	38610.
Baggage, Cargo	[lb]	10000.	10390.
Zero Fuel Weight	[lb]	362844.	364924.
Mission Fuel	[lb]	387156.	389554.
Take Off Gross Wght	[lb]	750000.	754478.

Table 1: Data for the Lockheed SST

The next step in the verification process consisted in the creation of a structural model together with load cases and material information. The wing and fuselage are mainly made of Ti6-Al-4V. Reference 12 and 13 were used to determine the material data at elevated temperatures. For the static load cases, material allowables were adjusted for a skin temperature of 250^o C which is roughly the temperature the hottest areas of the wing box will be exposed to during flight. An additional safety factor of 1.2 was then placed on the resulting allowables. Table 2 shows the material data of the titanium alloy Ti6-Al-4V used for the structural analysis.

	Lockh SST	HSCT	
Material	Ti6-Al-4V	Ti6-Al-4V	"Adv. Al."
Des. Temp. [^{o}C]	200	170	170
E [10^6psi]	16.0	16.0	12.0
ν	0.290	0.290	0.318
ρ [lb/in^3]	0.160	0.160	0.10
σ_x σ_y [psi] yield	82195	86700	61500
σ_{xy} [psi] yield	48458	50700	23300
ε_x, ε_y	0.00542	0.00542	0.00513
ε_{xy}	0.00817	0.00817	0.00482
safety factor	1.2	1.2	1.2

Table 2: Material Data

Lockheed studied a number of load cases including pull-up, pushover, landing, and flutter cases for different load conditions, but only a relatively small number of these cases proved to be the design drivers. For the sake of simplicity these were chosen as design load cases for the structural optimization performed in this research. Table 3 shows the load cases selected for the structural optimization, and Figure 5 depicts the flight envelope of the SST configuration with the design load cases marked. Static 2.5 g pull-up maneuvers were used along the edge points of the design dive speed envelope at full payload and full fuel tanks. A flutter margin of 1.2 on top of the dive speed was used in the Lockheed studies, therefore this margin was also applied here.

Fig. 7: ASTROS Convergence Histories, Lockh. SST

A modal and flutter analysis was performed with both final designs. The modal behavior of both cases and also the flutter behavior of the strength-only design were very similar to the Lockheed study. Table

4 shows a comparison of the lowest frequencies of the Lockheed study compared to the ASTROS results. The wing frequencies match very well, but even the fuselage bending is captured relatively accurately, although only a simple beam model was used to represent the fuselage.

	Frequency [Hz]	
Mode Description	Lockheed	ASTROS
Wing First Bending	0.915	0.976
Fuselage First Bending	1.345	1.104
Engine Pitch in Phase	1.494	1.973
Engine Pitch out of Phase	1.735	-
Fuselage Second Bending	2.478	2.096
Wing First Torsion	3.174	3.053

Table 4: Comparison of the Lowest Symmetric Modes, Lockheed SST

Conclusions of the Verification

The results obtained for the Lockheed SST model clearly demonstrate the validity of the approach. First and foremost, the structural models provide results that show very good agreement with the global data of the test cases. For the Lockheed SST, both the total designed weight and the static and dynamic behavior of the wing are exactly on target.

At the system level, it can be observed that FLOPS is applicable to supersonic transport configurations, but some scaling factors have to be used in order to obtain reasonable data. This applies mainly to the wing weight routine. Lift-dependent drag tables were supplied externally, therefore no statement can be made about FLOPS in that respect.

After both the Lockheed SST model and the model of the Concorde have produced satisfactory results both on the system level synthesis and in the structural design with ASTROS and PASCO, the multilevel decomposition procedure appears ready to be used for its original purpose, the HSCT wing design.

Design of a High-Speed Civil Transport Aircraft

Approach to the HSCT Design

The following approach was used for the HSCT design studies: at first, a system level baseline configuration was established. With this baseline aircraft, an initial structural optimization of the wing box was performed in order to obtain a realistic weight of the wing structure. A system level reanalysis with the actual weight of the wing structure then supplied the starting point for the actual design optimization. After each full design cycle (system level optimization - structural optimization), a review of the current design was performed in order to select the active design variables, design variable bounds, and wing box structural configuration for the next cycle. Once all levels had converged, the final design was thoroughly investigated and the results post-processed.

HSCT Baseline Configuration and Mission Profile

In order to be able to analyze and optimize different HSCT wing configurations, a baseline aircraft and a baseline mission were defined. Since the most lucrative market segment for the aircraft will be the transpacific traffic, it was felt that it is essential that the HSCT baseline is able to capture the largest part possible of this market. Although a range requirement of 6500 nm seems very challenging, it was used as the baseline range for these investigations. Current HSCT studies focus on an airplane seating capacity in the range of 250 to 300, where an aircraft at the upper range limit favorably having the lower number of seats, both due to the desire to keep the gross weight within reasonable limits and due to the fact that some of the longer routes do not have the passenger numbers to warrant a larger aircraft for at least one daily frequency. Therefore, the baseline HSCT was to have a 250 seat capacity and a design range of 6500 nm. Information from the NASA HiSAIR project[14,15,16] provided most of the remaining data needed for the baseline aircraft. A cruise Mach number of 2.4 was chosen. At this Mach number, the block time from the US west coast to Japan, the largest individual segment in terms of passengers, would be roughly 4.0 (Seattle - Tokyo) to 4.5 hours (Los Angeles - Tokyo), enabling the airplane to fly two daily round-trips with a turnaround time of 1.5 to 2.0 hours. Although a Mach number of 2.4 poses additional challenges for the HSCT in terms of high-temperature materials and cooling requirements, it was felt that two daily round trip flights for the primary market segment would be crucial to ensure a high aircraft utilization and thus economic success for the airlines.

The 9000 ft^2 wing thus obtained (Fig. 8) has an aspect ratio of 2.678 and a leading edge sweep of 73^o inboard and 43^o outboard. The resulting wing span is 155.25 ft. Information about the wing thickness and airfoil was not available, so a 3% thick airfoil was assumed, and for all the configurations analyzed an actual NACA 62003 airfoil was used as an envelope for the wing box. For this wing planform, the lift-dependent drag polars were determined with WINGDES[17], optimizing twist and camber for the initial cruise condition (Mach 2.4 and corresponding Reynolds number for 56,000 ft).

The baseline mission specified for the calculations consists of 10 minutes taxi and warm-up, take-off at sea level, standard day, climbout at 250 Kts TAS, accelerating climb to the initial cruise altitude of 56,000 ft, then a supersonic cruise at Mach 2.4 and optimum altitude for maximum specific range to the destination. After descent, landing and taxi for 5 minutes, standard reserves for a flight for 250 nm to an

Fig. 10: HSCT Flight Envelope

Initial Design Cycle

With this structural model, five design iterations of the ASTROS-PASCO procedure were performed. The final designed weight is 27,445.6 lb for one wing, a value that seems pretty much converged, as the design history (Fig. 11) shows. This leads to a

Fig. 11: Design History for ASTROS-PASCO, Initial Design Cycle

In the subsequent design review it was discovered that a translation of the relative wing break point location from 41% span to 50% span led to both aerodynamic and structural benefits. Since this parameter is not currently a FLOPS design variable, a trade study had to be performed, varying the breakpoint location by hand and then fixing it at its "best" value. This value of 50% was also used in the structural optimization loop. Otherwise the wing structure remained unchanged. Also at this point, a trade study was performed, investigating the influence of an increased number of spars and ribs, different material configurations, wing sweep and thickness-to-chord ratio on the structural weight. Although the influence of each individual parameter was relatively small, it was felt that a combination of all "good" characteristics could lead to an additional improvement. For details, see ref. 11.

		Conv. Baseline	Final Config.
Performance, General			
Passengers		250	250
Range	[Nm]	6500	6500
Cruise Mach No.		2.4	2.5
Max. Cruise Altitude	[ft]	70000	70000
Productivity Index	[Kts]	89.93	103.45
Engine Type		Turbine-Bypass	Turbine-Bypass
Net Thrust (SLS)	[lb]	50000	42495.3
Number of Engines		4	4
Geometric Data			
Fuselage Length	[ft]	280.0	280.0
Wing Span	[ft]	155.25	133.68
Wing Area	[ft^2]	9000.0	8491.3
Aspect Ratio		2.678	2.103
LE Sweep, Inboard	[deg.]	73.0	73.0
LE Sweep, Outboard	[deg.]	43.0	41.22
Wing T / C Ratio	[%]	3.0	3.0
Weights			
Wing	[lb]	102300.	73275.
Fuselage	[lb]	36593.	36593.
Total Structure	[lb]	162925.	131891.
Propulsion	[lb]	58136.	49661.
Syst. and Equipm.	[lb]	39059.	38830.
Empty Weight	[lb]	260120.	220382.
Operating Items	[lb]	7643.	7505.
Oper. Empty Weight	[lb]	267763.	227887.
Passengers	[lb]	41250.	41250.
Baggage, Cargo	[lb]	13545.	13545.
Zero Fuel Weight	[lb]	322558.	282682.
Mission Fuel	[lb]	452875.	418730.
Take Off Gross Wght	[lb]	775433.	701412.

Table 6: Main Characteristics of the FLOPS-ASTROS-PASCO HSCT Baseline Configuration

For the next design cycle, cycle II, the wing structural weight was passed up to the system level, and the design variables Mach number and wing sweep were activated in addition to those of the previous cycle. Again Fig. 12 displays the system level results of this design cycle. The productivity index was further increased to a value of 101.07 Kts, and 102.09 Kts after correction of the aerodynamic tables at the end of the optimization. A large portion of the increase in PI is

I.2 I.4 II.1 II.2 II.4 II.6 II.8

Fig 12: System Level Design History

At this point, all features determined as "good" in the structural trade study after the previous cycle were combined, i.e. increased number of ribs to reduce the buckling length and exclusively titanium as the material. The flight envelope was extended to account for the increased cruise Mach number of 2.5. With this structural configuration the structural optimization was run to full convergence which it reached after seven iterations, further reducing the designed wing structure weight to 20351.6 lb for one wing.

This value was passed back up to the system level, leading, including follow-up effects, to a reduced gross weight of 701,412 lb and a PI of 103.45 Kts, starting point for the next design cycle. The subsequent system level optimization did not move in the design space anymore, therefore this design was considered the final, optimized design. Table 6 shows the main characteristics of the final in comparison with the initial configuration with a wing area of 8491 ft^2 and a total wing weight of 73275 lb.

The Mach 2.5 configuration described here was just one of the designs analyzed in detail, but it is very much representative for the studies performed. For additional information, please refer to ref. 11.

Conclusions and Outlook

The main objective of this thesis research was twofold, on the one hand the development of a multidisciplinary design environment to demonstrate how a system can be decomposed to subsystems and down to the component level, and how it is then recomposed and the detailed component and subsystem information is processed on the system level, and on the other hand the application of this procedure to a very challenging research area, the design of a second generation supersonic transport aircraft.

The multidisciplinary framework has been established and verified with two sample cases, the results of which are very convincing. The environment is modular and flexible and open for the addition of further disciplinary modules. As a side product, a powerful finite element model generator has been developed that rapidly creates complete wing box finite element models based on geometry and mission parameters from a design synthesis program. The model generator is applicable to any type of wing from gliders to fighter aircraft to large subsonic, and, last but not least, supersonic transport configurations.

The calculation of the wing weight from the finite element model weight multiplied by a scaling factor determined with the help of existing similar wings is not the best solution. A better procedure that may be available in the future simply affects the actual values transferred to the system level tool, not the principle of supplying an accurate finite element model weight as such, therefore can be easily substituted.

With respect to the second focus of the work, the HSCT design, the optimum configuration - within the margins of error of preliminary design - has been determined based on a very detailed structural analysis and less sophisticated methods for the other disciplines. A comparatively sophisticated aerodynamic tool, coupled with FLOPS, will be the next logical step to further improve the design - or at least increase the level of confidence in the present design. With this comprehensive MDO framework in place, it is now possible to analyze the impact of new technologies in terms of materials, aerodynamics, and propulsion on the overall HSCT design. As the studies have shown, no very significant improvement over the baseline configuration was possible with current technology levels in all areas.

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